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Prediction of the Rear Fuselage Temperature with Radiation Shield

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***Abstract*—**In order to enhance the aircraft survivability, the infrared signatures emitted by hot engine parts should be determined exactly. For its reduction it is necessary for the rear fuselage temperature to be decreased. In this study, numerical modeling of flow fields and heat transfer characteristics of an aircraft nozzle is performed and its temperature distribution along each component wall is predicted. The radiation shield is expected to reduce the skin temperature of rear fuselage. The effect of material characteristic of radiation shield on the heat transfer is also investigated. Through this numerical analysis, design parameters related to the susceptibility of aircraft are examined.

***Keywords*—**Infrared signature, Nozzle flow, Radiation shield, Rear fuselage temperature, Susceptibility

I. Introduction

SURVIVABILITY of aircraft has been a one of the important factors in modern battlefield in which the standard of technology determines the supremacy. Military forces are engaged in equipping a stealth function to the fighter plane at the early stage of design. The importance of susceptibility problem has been strengthened related to detection and tracking by enemy missiles or seekers. It is well known that aircraft survivability is threatened by detection using radar and thermal infrared(IR) sensor as well as by primary ways such as visual and sound sources. Several varieties of detection systems have been developed and their technology has been more sensitive, therefore signature reduction technology is excessively in need of development. Advanced countries lead the survivability and susceptibility research [1, 2], but they strongly control the outflow of these technologies overseas.

High level of infrared signature emitted by aircraft rear fuselage is an easy target for IR detectors, therefore reduction of IR signature is essential for the improvement of survivability. IR signature can be predicted by figuring out the distribution of the rear fuselage temperature, because IR signature is a function of temperature. The numerical analysis of nozzle flow needs to be preceded to estimate the skin temperature precisely, and the factor that affects the rear fuselage temperature can be identified using the results. In the following, the mathematical formulation and the corresponding numerical schemes are introduced with radiation shield, and then parametric studies are performed by changing shield and material characteristics for the prediction of aircraft nozzle flow and rear fuselage temperature. Finally, some concluding remarks are presented.

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II. Nozzle Flow Analysis

A. Nozzle Flow

Fig. 1 illustrates the engine layout examined in this paper. It consists of jet nozzle, radiation shield, and casing. The convergent-divergent nozzle shape [3] is considered for the supersonic flow, and the outermost duct is engine casing exposed to the freestream. The radiation shield and engine casing are both considered a thin shroud. The radiation shield is set between nozzle wall and outer casing to cut off radiation heat, which is generally used to suppress radiative heat transfer in need of insulation. Radiative heat can be effectively diminished by placing more than one thin metal shroud between surfaces. It can be applied to aircraft rear fuselage to reduce the heat from nozzle to the outside [4]. Hot combustion gas (FLUID 1) flows through the nozzle, and cooling air (FLUID 2) passes through inside both the radiation shield and the casing. Outside of engine casing, the freestream flow (FLUID 3) exists, which shows the atmospheric condition and the flight speed of aircraft. To simulate the flow fields of rear fuselage, the computational domain is topologically divided into 4 blocks, as sketched in Fig. 1.

In case of using jet fuel with molecular formula $C_{11}H_{22}$, the flow inside the nozzle can be assumed to be composed of about 13% carbon dioxide, 13% water vapor, and 74% nitrogen. The flow becomes high temperature and high pressure condition passing through the combustor and turbine. In this study, the total pressure and total temperature for nozzle inlet is set to 3.41

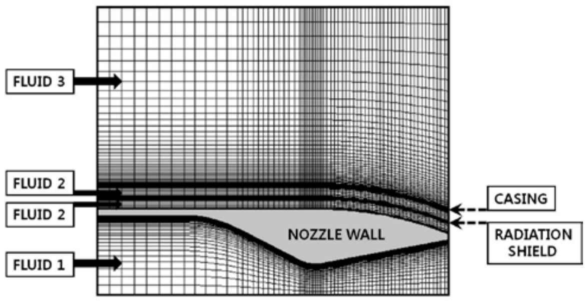

Fig. 1 Schematic of engine layout

atm and 2000 K, respectively. The freestream mach number is assumed to be 2.05. The condition of each flow field is as follows:

FLUID 1: $P_T = 3.41\, atm\, , T_T = 2000\, K$

FLUID 2: $P = 1\, atm\, , T = 300\, K\, , M = 0.3$

FLUID 3: $P = 1\, atm\, , T = 300\, K\, , M = 2.05$

B. Boundary Condition

In order to predict the nozzle wall temperature precisely, heat transfer between gas and nozzle wall and heat conduction in a solid wall should be analyzed simultaneously. At the wall surface, heat is transferred by conduction and radiation and the heat balance can be described by following equation:

$$q_{c,g} + q_{r,g} = q_{c,s} \tag{1}$$

As the radiation shield and outer casing are both considered as a thin shroud, heat transfer by the inner and outer flow field should be balanced on each wall. Therefore, the boundary condition for each wall can be defined as

Nozzle inside wall: $q_{c,F1} + q_{r,F1} = q_{c,wall}$

Nozzle outside wall : $q_{c,F2} + q_{r,F2} = q_{c,wall}$

Radiation shield: $q_{in,c,F2} + q_{in,r,F2} = q_{out,c,F2} + q_{out,r,F2}$

Casing: $q_{c,F2} + q_{r,F2} = q_{c,F3} + q_{r,F3}$

where FLUID 2 flows on both the inner and outer side of the radiation shield. Heat transfer from the inner side to the wall is written by q_{in} and one from the outer side to the wall is written by q_{out} to avoid confusion.

III. Numerical Formulation

A numerical simulation was modeled to predict the skin temperature of aircraft rear fuselage. To analyze the nozzle flow field, the preconditioned Navier-Stokes scheme is employed to be applicable to flows at all mach numbers. To calculate the inviscid flux, the AUSM$^+$-up scheme is adopted.

A. Preconditioning Algorithm

Inside the nozzle, the subsonic and supersonic flows coexist and the convergence is decreased due to the difference of compressibility. The preconditioning scheme is applied to solve effectively both incompressible and compressible area and in this paper, the preconditioning matrix defined by Weiss and Smith [5] was taken. The preconditioned two-dimensional axisymmetric Navier-Stokes equations with pressure, velocity, and temperature (p, u, v, T) as primitive variables are expressed as

$$\Gamma \frac{\partial Q_v}{\partial t} + \frac{\partial (E - E_v)}{\partial x} + \frac{\partial (F - F_v)}{\partial y} = H_s \tag{2}$$

where Γ is the preconditioning matrix, Q_v is the vector of primitive variables, E and F are the inviscid fluxes, E_v and F_v are the viscous fluxes, and H_s is an axisymmetric vector. Γ is defined as

$$\Gamma = \begin{pmatrix} \Theta & 0 & 0 & \rho_T & 0 & 0 & 0 \\ \Theta u & \rho & 0 & \rho_T u & 0 & 0 & 0 \\ \Theta v & 0 & \rho & \rho_T v & 0 & 0 & 0 \\ \Theta H - 1 & \rho u & \rho v & \rho_T H + \rho C_p & 0 & 0 & 0 \\ \Theta \kappa & 0 & 0 & \rho_T \kappa & \rho & 0 & 0 \\ \Theta \omega & 0 & 0 & \rho_T \omega & 0 & \rho & 0 \\ \Theta Y_i & 0 & 0 & \rho_T Y_i & 0 & 0 & \rho \end{pmatrix} \tag{2a}$$

where,

$$\Theta = \frac{1}{\theta} - \frac{1}{a^2} + \frac{1}{RT} \tag{2b}$$

$$\theta = min\left[a, max\left(V, V_{free} \times 0.5\right)\right]^2 \tag{2c}$$

B. AUSM$^+$-up Scheme

AUSM$^+$-up scheme, one of the modified AUSM-family (Advection Upstream Splitting Method) schemes, is applied to approximate the inviscid fluxes in the Navier-Stokes equations. AUSM scheme has the advantage to provide a crisp resolution of strong shocks and accurate results for boundary layers, and AUSM$^+$-up scheme has been developed by Liou [6] to be reliable over the entire speed regimes.

$$E, F = \dot{m}_{1/2} \begin{cases} \vec{\phi}_L \\ \vec{\phi}_R \end{cases} + P_{1/2} \tag{3}$$

where,

$$\vec{\phi} = (1, u, v, H)^T \tag{3a}$$

$$\dot{m}_{1/2} = u_{1/2}\, \rho_{1/2} = a_{1/2} M_{1/2} \begin{cases} \rho_L & if\ u_{1/2} > 0 \\ \rho_R & otherwise \end{cases} \tag{3b}$$

$$M_{1/2} = M^+_{(4)}(M_L) + M^-_{(4)}(M_R) + M_p \tag{3c}$$

$$P_{1/2} = P^+_{(5)}(M_L) P_L + P^-_{(5)}(M_R) P_R + P_u \tag{3d}$$

IV. Results and Discussion

We have developed the nozzle thermal flow analysis code based on numerical techniques explained previously, which was verified by comparing with the experimental data in the previous work [7], therefore, it is not repeated here.

The condition of each flow field was described in section Ⅱ. Multi-block grid system is adopted and the freestream region extends out from the outermost duct to about 3 radii in the radial direction. Here, the heat conductivity of nozzle wall is set to 20 W/(m·K).

Fig. 2 shows the resulting Mach number distribution. As the gas flows through the nozzle, the flow is accelerated and becomes supersonic. As illustrated in Fig. 2, at the throat, the flow near the nozzle wall has a higher Mach number than near the centerline. This is because pressure expansion passing through the throat region occurs rapidly near the wall than nozzle axis. At the freestream domain, it can be seen that an expansion wave is created along the nozzle wall shape and the Mach number increases as the pressure decreases.

A. Effect of the Radiation Shield

The temperature profile of each wall is described in Fig. 3. First of all, for the case of without radiation shield, which is depicted in solid line, as the hot combustion gas enters the nozzle inlet, the heat is transferred outward and the temperature of nozzle outside wall and outer casing increases. When the flow accelerates inside the nozzle the pressure and temperature drop, therefore the outer casing temperature decreases along the flow direction. The cooling air inside the casing also affects this temperature drop. In the nozzle inner wall, unlike the nozzle outside wall and casing, it can be seen that the evolution of a temperature profile is changed between the starting point of the converging section and the throat. It can be deduced that this phenomenon occurs because of the change of nozzle wall thickness. Since the nozzle wall acts like a resistance in heat transfer, thus as the thickness of nozzle wall increases, the amount of heat getting out is reduced and the temperature of the inside wall rises locally. However, there is a point where the temperature diminishes suddenly where the curvature of nozzle throat begins. As mentioned earlier, the pressure near the wall falls rapidly and the temperature also drops.

Now, we compare the resulting temperature distributions with and without the radiation shield. As shown in Fig. 3, even if the outer casing places the same location, its temperature profile shows the difference depending on the domain of the radiation shield. For the case without a shield (solid line in figure), casing has a temperature range of about 550 K to 950 K, whereas the one with a shield (dotted line in figure) ranges from 370 K to 640 K. By placing the radiation shield between nozzle wall and outer casing, the shroud blocks out the heat partially, and as a result, we can get an effect on reducing rear fuselage temperature by 26 to 40 percent.

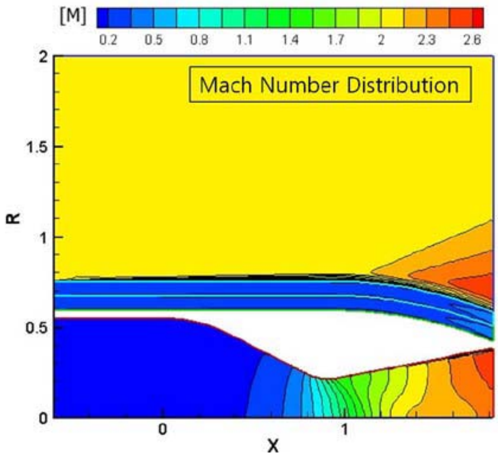

Fig. 2 Contour plot of the Mach number

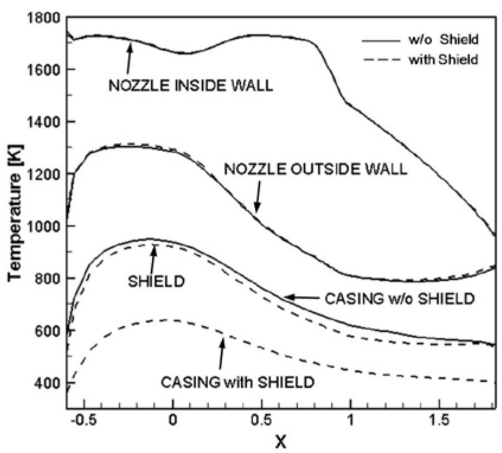

Fig. 3 Effect of the radiation shield on temperature

B. Effect of Material Characteristic

In this section, effects of material characteristics on temperature of nozzle wall, radiation shield, and casing of the rear fuselage are examined. For this, the thickness of the radiation shield and casing should be considered as well as the nozzle wall. The thickness of shield and casing is set to be same as that of nozzle wall inlet as illustrated in Fig. 4.

In previous section, as we assumed that the radiation shield and casing are thin shrouds, only the heat conductivity of nozzle wall was considered. However, now we consider the solid conduction in all 3 walls and set the heat conductivity of each wall to 20 W/(m·K). In fact, thermal heat conductivity is a function of temperature, in this work, however, it is assumed to be constant. Fig. 5 shows the temperature distribution when the all walls have thickness. Compared to Fig. 3, the tendency of temperature profile of each wall is same regardless of thickness.

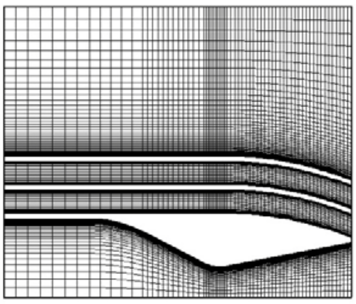
Fig. 4 Thickness of shield and casing

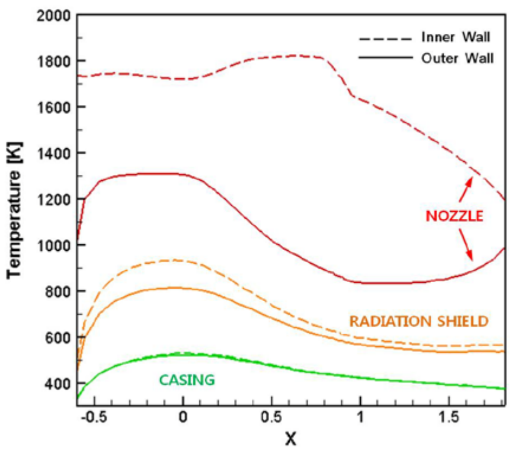

Fig. 5 Thickness effect of shield and casing

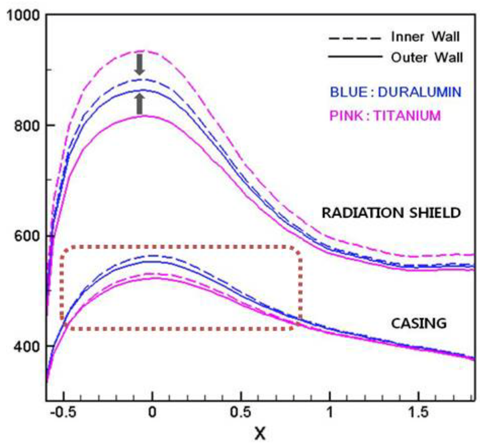

Fig. 6 Effect of conductivity of the radiation shield

As a natural result, both radiation shield and casing have different inside and outside temperature. For the casing wall, the temperature difference between inside and outside wall is not as significant as that of shield, because the temperature is low enough compared with outer ambient temperature. Through this result, we can suggest that the material characteristic of radiation shield is more important than that of outer casing. Therefore, we next examine the effect of thermal conductivity value of radiation shield on the temperature.

Maintaining the value of conductivity of nozzle wall and casing by 20 W/(m·K), we have just changed the value of radiation shield. Materials used for an aircraft fuselage need to be strong, hard, and lightweight, and titanium, duralumin, and so on belong to those materials. Figure 6 shows the temperature profiles when the radiation shield is considered to be made of titanium or duralumin. Thermal conductivity of titanium and duralumin are 20 W/(m·K) and 141 W/(m·K), respectively. As the conductivity increases, more heat is transferred through the solid, thus the temperature of inside wall becomes lower and that of outside wall becomes higher. As shown in Fig. 6, for duralumin case, the shield transfers more heat and outside wall temperature of shield increases compared to titanium case. Because the outer casing is affected by shield temperature, it has been observed that the casing temperature for duralumin shield is higher than that for titanium shield. That is, to reduce the temperature of outer duct, low-conductivity metal should be chosen as radiation shield material.

V. Conclusion

In this study, we examined the design parameter which can reduce the IR signature of aircraft by performing the analysis of nozzle thermal flow. As expected, since the IR signature is related to the temperature, decreasing the rear fuselage temperature is crucial.

Rear fuselage temperature can be reduced by placing the radiation shield between nozzle wall and casing, because the shield blocks the radiative heat transfer effectively. The material characteristics of rear fuselage also affect the wall temperature. Material with low conductivity transfers less heat through the wall, thus outside wall temperature is decreased and consequently the temperature of outer casing wall is decreased.

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